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ANALYSIS OF FACTORS AFFECTING NET LIFT INCREMENT ATTAINABLE WITH TRAILING-EDGE SPLIT FLAPS ON TAILLESS AIRPLANES

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ADVANCE RESTRICTED REPORT

ANALYSIS OF FACTORS AFFECTING NET LIFT INCREMENT ATTAINABLE WITH TRAILING-EDGE SPLIT FLAPS ON TAILLESS AIRPLANES

By Marvin Pitkin and Bernard Maggin

SUMMARY

An analysis has been made of factors affecting the net lift increment attainable with trailing-edge split flaps on tailless airplanes. The flaps investigated in the analysis were designed to contribute zero pitching moments about the wing aerodynamic center when deflected. Calculations were made of the lift and pitching-moment characteristics of flaps of this type over a range of design conditions in which sweepback angle, aspect ratio, taper ratio, flap chord, and flap deflection were widely varied. In addition, calculations were made to determine the effect of the various parameters upon the loss in lift incurred in trimming the stability moments of a tailless airplane. A method is given for roughly estimating the maximum lift coefficient of tailless airplanes.

The results of the analysis indicated that aspect ratio and sweepback angle were the principal parameters influencing the net lift increment attainable with the flaps on tailless airplanes in trimmed flight. An increase of these parameters allowed the use of larger-span flaps. Large values of both parameters were required to obtain sizable lift increments and to minimize the loss of lift caused by the longitudinal control surfaces. In order to utilize fully the high-lift advantages associated with flaps on swept-back wings, the use of tip slots or washout will probably be required to eliminate the undesirable tip stalling and the accompanying stability losses caused by large angles of sweepback.

The allowable flap span - and hence the net lift increase - of some wing configurations could be further increased by the addition of a trim flap located at the wing tips and deflected upward.

Excessive wing taper was shown to reduce the net lift increment obtainable from flap systems, whereas increased flap chord and deflection increased the net lift increment.

INTRODUCTION

The application of high-lift flaps to a tailless airplane requires a flap arrangement that produces only small pitching moments about the center of gravity of the airplane. Such an arrangement is necessary because the elevators on tailless airplanes operate on short moment arms and thus produce relatively small pitching moments.

The pitching moments produced by flaps may be kept small by the use of a basic flap design that has a small section pitching moment or by the use of partial-span flaps on wings with sweepback. Another method of reducing the pitching moments of flaps is to cancel out the diving moments of one flap deflected downward (designated lift flap) by means of a second flap deflected upward (designated trim flap) and possessing a longer lever arm. An example of this method of obtaining increased lift by means of such a multiple-flap system is shown by the conventional airplane in which the trim flap (elevator) possesses a moment arm from 10 to 15 times as long as that of the wing lift flaps.

Much information is available on the section pitching moment and lift produced by various flap designs, and some work has been done on methods of computing complete-wing moments from section data although the data have not been directly applied to tailless airplanes. In the present report, the results of an analytical investigation are given for a wide range of wing and flap parameters. The effect of these parameters upon the net lift increment obtainable from flaps on tailless airplanes in trimmed flight has been treated. Conventional split flaps were chosen for the investigation because of their simplicity and because they produce relatively small section pitching moments for a given lift increase.

Calculations were made to determine the effect of sweepback, aspect ratio, taper ratio, flap chord, and flap deflection upon the lift increment obtainable at a

fixed angle of attack by means of flaps creating zero pitching moments about the wing aerodynamic center. Additional calculations were made to determine the losses in lift resulting from the elevator deflections required to trim stability moments. A brief study was made of available empirical data concerning the effect of sweepback and flaps upon the maximum lift probable in trimmed flight.

The lift increments of different flap arrangements were determined by the method of reference 1 with a simple chord correction factor being applied. The pitching moments were computed from a consideration of the incremental lift due to the action of the flaps at each spanwise section and the center of pressure of this incremental lift. This procedure is somewhat similar in basic principles to the method of reference 2. The accuracy of the present method was determined by calculating by means of this method the lift and moment increments of flaps on 10 different finite wing-flap combinations for which wind-tunnel data were available.

SYMBOLS

α angle of attack, degrees

C_L lift coefficient $\left(\frac{\text{Lift}}{qS}\right)$

$C_{L_{max}}$ maximum lift coefficient

c_l section lift coefficient $\left(\frac{\text{Section lift}}{qc}\right)$

$\left(C_{L_{max}}\right)_{tr}$ maximum lift coefficient in trimmed flight

ΔC_L increment of lift coefficient

C_m pitching-moment coefficient $\left(\frac{\text{Pitching moment}}{q\bar{c}S}\right)$

c_m section pitching-moment coefficient $\left(\frac{\text{Section pitching moment}}{qc^2}\right)$

ΔC_m	increment of pitching-moment coefficient
q	dynamic pressure $\left(\frac{1}{2}\rho V^2\right)$
ρ	density of air, slugs per cubic feet
S	wing area, square feet
V	true airspeed, feet per second
b	wing span, except as designated otherwise by subscript, feet
c	wing chord at any section, except as designated otherwise by subscript, feet
c_s	root chord, feet
$\bar{c}$	mean wing chord, feet (S/b)
$c.p._f$	center of pressure of incremental lift load caused by flap deflection, fraction of wing chord
x	distance from center of pressure to reference point, feet
t/c	airfoil thickness, fraction wing chord
$\Delta\alpha$	equivalent change in angle of attack for a given flap deflection, degrees
A	aspect ratio (b^2/S)
λ	taper ratio; ratio of tip chord of wing to root chord of wing
Λ	angle of sweepback of quarter-chord line, degrees
F, H	theoretical factors that are functions of aspect ratio and taper ratio
h	static-margin factor $\left(\partial C_{m_{c.g.}}/\partial C_L\right)$

$x_{a.c.}$	distance to aerodynamic center of wing from quarter chord of root section, feet
$c/4$	quarter-chord line
δ	control-surface deflection, degrees
a	slope of lift curve, per degree $(\partial C_L/\partial \alpha)$

Subscripts:

f	flap
L	lift flap
T	trim flap
c.g.	about center of gravity
a.c.	about wing aerodynamic center
o	infinite aspect ratio
theor	pertaining to theoretical wing plan forms given in reference 1
max	maximum
1	over flapped part of wing
2	over unflapped part of wing
p	longitudinal (pitch) control
w	unflapped finite wing
bw	basic wing with zero sweepback of quarter-chord line

METHODS

Calculation of Incremental Lift Caused by Flaps

The incremental lift caused by flap deflection at a constant angle of attack was obtained by integration of the section incremental load distribution across the

wing span. These span load distributions were calculated by the influence-lines method described in reference 1. Inasmuch as the data in reference 1 apply rigorously only to the wing shapes shown in that report, a chord correction was applied to the span load distributions obtained from such data. No correction was applied to the span loading to account for the effect of sweepback, however, because available data on the subject were inconclusive. The span loading was corrected for chord by multiplying the loading at each spanwise station of the wing shape most similar to that under consideration by the ratio of the chord of the wing under consideration to the chord of the wing shape of reference 1.

The method of reference 1 gives the value of incremental lift caused by flaps that create an effective change in angle of attack of 1 radian over the flapped parts of the wing. In order to convert such data to the incremental lift caused by a different flap deflection, the equivalent increment of angle of attack $\Delta\alpha$ for that deflection had to be found. For the present analysis, values of $\Delta\alpha$ were obtained from data for wings of infinite span at a lift coefficient of 1.0, which corresponds to an angle of attack of 10° for the average unflapped wing section. This value was chosen because the angle is far enough below the stall to yield consistent results and, at the same time, is in the high-lift range at which flaps are utilized. A typical incremental span load distribution caused by flap deflection is presented in figure 1(a).

The values of $\Delta\alpha$ used in the analysis were obtained from reference 3 for split flaps of various chords and are plotted against flap deflection in figure 2. The lift increments of constant-chord flaps or arbitrary-chord distributions were calculated by use of a value of $\Delta\alpha$ based on the mean flap chord.

Calculation of Pitching Moments Caused by Flaps

The incremental pitching moments caused by the flaps were calculated by multiplying the incremental lift loads caused by flaps at each spanwise station by the corresponding moment arms - that is, the distance between the local center of pressure of the lift load and the moment axis. Graphical integration of the spanwise distributions of pitching moments thus obtained yielded the incremental pitching moments created by flap deflections.

<u>Calculation of centers of pressure of flap loads over flapped parts of wing</u>.- The center of pressure of the incremental lift loads over the flapped parts of the wing $c.p._{f_1}$ was calculated from section data by the simple relation

$$c.p._{f_1} = \frac{a_o}{a}\left(c.p._{f_o} - 0.25\right) + 0.25 \qquad (1)$$

Equation (1) can be rigorously applied only to a finite wing that is equipped with full-span flaps for which the local lift-curve slopes are equal to that of the wing as a whole. This equation, however, was found to define the center of pressure of the flap load over any flapped part of the wing with satisfactory accuracy.

Values of the lift-curve slope for wings of infinite aspect ratio a_o vary with airfoil thickness and were selected from the data of reference 2. Values of the lift-curve slope for finite wings a were calculated by the formula

$$a = \frac{Fa_o}{1 + \frac{57.3a_o}{\pi A}} \qquad (2)$$

also presented in reference 2. The factor F, which is a correction factor involving aspect ratio and taper ratio, varies between 0.98 and 1.00. A value of 0.99 was used in the present report.

A study of data for the NACA 230-series airfoil (reference 3) indicated that the center of pressure of incremental flap loads on wings of infinite aspect ratio $c.p._{f_o}$ was independent of flap deflection but varied somewhat with angle of attack and considerably with airfoil thickness and flap chord. All values of $c.p._{f_o}$ were chosen at an angle of attack of 10°, the angle of attack at which the lift increments were calculated. The effect of flap chord and airfoil thickness upon $c.p._{f_o}$ was determined from the force-test data of reference 3 and is shown in figure 3. The variation of $c.p._{f_o}$ with

flap chord, as calculated by lifting-line theory for wings of infinite aspect ratio, is also given in figure 3 for comparison.

By use of equation (1), calculations were made to determine the center of pressure of the incremental flap loads over the flapped parts of the finite wing $c.p._{f_1}$. Figure 4 presents the calculated variations of $c.p._{f_1}$ with aspect ratio for split flaps of various chords on moderately thick (16 percent) airfoils. Agreement of these calculated values with available data for rectangular wings with full-span flaps was found to be good.

Center of pressure of incremental flap loads over unflapped parts of the wing.- The center-of-pressure distribution of the incremental flap loads over the unflapped parts of the wing $c.p._{f_2}$ was obtained from the pressure-distribution data of reference 6 that are shown in figure 5. These data indicated that the value of $c.p._{f_1}$ may be satisfactorily calculated by equation (1) and that the spanwise distribution of $c.p._{f_2}$ may be simulated by a line faired between the 0.40c station at the flap end to the 0.25c station at a point $0.30\frac{b}{2}$ from the flap end; the induced loads due to the flap may thereafter be considered to act along the 0.25c line. By use of the values thus obtained, the center-of-pressure distribution of incremental loads caused by flaps may be readily determined for any desired wing-flap configuration. Figure 1(b) illustrates typical center-of-pressure distributions for constant-chord flaps. The pitching-moment distributions obtained by use of such data in conjunction with the span loading are illustrated in figure 1(c).

Accuracy of Methods

In order to determine the over-all accuracy of the methods and data presented herein, the incremental flap lift and pitching-moment coefficients were calculated for 10 finite wing-flap combinations for which wind-tunnel force-test data were available. The close agreement of calculated with measured values is shown in figure 6.

Lift Evaluation under Trimmed-Flight Conditions

A certain percentage of the lift created by the lifting surfaces of an airplane is lost in trimmed flight owing to the action of the trimming surfaces in balancing pitching moments. Such a loss of lift is considered herein to be divisible into two parts. The first part is the loss of lift that is encountered in trimming flap moments; the second part is the loss encountered in trimming the pitching moments created by the longitudinal stability of the airplane. These lift losses have been treated separately in the present analysis.

Net lift increment caused by flaps.- If the basic-wing pitching moments at zero lift are neglected, the lift lost in trimming the flap moments is caused by the trimming surfaces in balancing the flap moments about the wing aerodynamic center. The net gain in lift caused by flap deflection is, therefore, the lift increment caused by flap deflection minus the lift lost in trimming the flap moments about the wing aerodynamic center. The net gain in lift caused by flaps in trimmed flight is therefore automatically obtained for the flaps that create zero pitching moments about the wing aerodynamic center. All flap combinations investigated in this analysis were, for this reason, designed to create zero incremental pitching moments about the wing aerodynamic center.

For each design configuration investigated, the incremental lift of the flaps was obtained for a series of lift flaps of various spans extending outboard from the center line of the wing and for a series of trim flaps extending inboard from the wing tip.

The location of the aerodynamic center for the wing shape under investigation was then determined from the following formula given in reference 2:

$$x_{a.c.} = Hb \tan \Lambda \tag{3}$$

where H is a function of aspect ratio and taper and may be obtained from reference 2.

After the location of the wing aerodynamic center has been determined, the incremental pitching moments of

the lift and trim flaps were calculated about the wing aerodynamic center. Each lift flap was then matched with a trim flap so that the combined incremental pitching moments of the two flaps were equal to zero. For some arrangements, a lift flap existed that created no moments about the aerodynamic center. This type of flap is referred to herein as a "self-trimming flap." The net lift increment used to evaluate a given flap system was then obtained by deducting the loss in lift due to the trim flap from the lift increment created by the corresponding lift flap.

Equation (3) does not take account of the rearward shift of aerodynamic center caused by the shift of loading toward the wing tip as the wing is swept back. The data of reference 10 and unpublished force-test data from the NACA 19-foot pressure tunnel indicate that this shift may be as large as 5 percent of the mean aerodynamic chord. The results of the analysis are therefore believed to be somewhat conservative and should probably indicate net lift increments larger than those actually determined.

Lift loss caused by longitudinal stability.- The lift lost by the longitudinal control surfaces in trimming stability moments is associated with a pitching moment about the wing aerodynamic center equal and opposite to hC_L, where h is the static-margin factor $\partial C_{m_{c.g.}}/\partial C_L$. This loss is zero when the longitudinal stability is zero ($h = 0$) and may therefore be attributed to the airplane design rather than to the flap characteristics.

Inasmuch as the longitudinal control surfaces of tailless airplanes are normally mounted at the wing tips, the data from the trim-flap calculations were used to obtain the lift lost by such controls for various stability moments.

SCOPE OF CALCULATIONS

All calculations made in the present analysis were based on values of flap effectiveness and the center of pressure of the flap load obtained by averaging section force-test data for the NACA 23012 and NACA 23021 airfoils. The calculations are therefore most applicable to airplanes equipped with moderately thick wing sections (about 16 percent) of the NACA 230 series.

Calculations were made first of the net lift increases, at a constant angle of attack, caused by 0.30c flaps deflected 60° and installed on a wing with aspect ratio of 7.3, sweepback angle of 20°, and taper ratio of 0.25. Each of these wing and flap parameters was then varied independently of the others. The design parameters were varied as follows:

Sweepback angle, degrees	10, 20, 30
Aspect ratio	6, 7.3, 10, 16
Taper ratio	0.25, 0.50, 1.00
Flap chord, percent c	10, 20, 30, 40
Flap deflection, degrees	0 to 60

The lift loss caused by trim flaps of various spans was also calculated over the range of design parameters in order to permit estimation of the lift lost by deflection of the longitudinal control surface. The maximum lift coefficient of a tailless airplane that incorporated the design features shown to be favorable for obtaining high lift was then estimated.

RESULTS AND DISCUSSION

The results of the analysis are presented in figures 7 to 14, which show the variation of the allowable flap span and the net lift increments due to flaps with each of the basic design parameters. A gain in the net lift increment obtained by the variation of one design parameter cannot be added directly to the gain obtained by the variation of another parameter because these gains were generally obtained by an extension in flap span. Two design changes, each of which permits an extension of a given flap equal to 0.40b would not therefore necessarily permit an extension of 80 percent when acting together. Such changes would result in a larger allowable flap span, but the actual quantitative magnitude of the increase would have to be recalculated.

Effect of Flap Chord

Lift-flap chord.- The effect of increasing the chord of the lift flap is shown in figure 7. Arrows and points are included as an aid in using the chart, and an example of the use of this figure follows:

If the chord and span of a self-trimming flap that yields $\left(\Delta C_{L_f}\right)_{net} = 0.31$ are to be found, point a is first located on the net-lift scale and the arrow is followed to point b. Point b indicates that a lift-flap chord of 0.235c is required. Following the arrows from point b to point c and then to point d indicates that the required flap extends over the inboard section a distance equal to 0.20b. In a similar manner, the dimensional characteristics of a multiple-flap system may be obtained. For example, following the arrows from point a to points e, f, and g shows that the same net incremental lift may be obtained from a system consisting of a 0.36b lift flap and a 0.20b trim flap, both of $0.13\bar{c}$.

The data presented in figure 7 show that an increase in flap chord on a tailless airplane causes an almost linear increase in the net lift increment due to the flaps. Although this lift increase is caused largely by the increase in $\Delta\alpha$ that accompanies an increase in flap chord, this action is further reinforced by the forward shift of the center of pressure of flap loads with increased flap chord. This center-of-pressure change allows extension of the flap span for a given flap pitching moment and hence increases the net lift attainable in trimmed flight.

The data of figure 7 show also that the addition of trim flaps to the outboard wing sections permits an extension of the allowable lift-flap span and results in increased net lift increments for a given pitching moment. The increase in lift with a given increase in flap chord is more pronounced for the multiple-flap systems because of the greater allowable span of the lift flaps. A multiple-flap system occupies more span for a given lift increment, however, than a self-trimming flap and consequently leaves less span for lateral and longitudinal control surfaces.

Trim-flap chord.- The effect of varying the chords of the trim flaps required to trim a series of constant-chord lift flaps is shown in figure 8. These data show that varying the trim-flap chord had no appreciable effect upon the net lift produced by a multiple-flap system but that the principal effect was to decrease the required trim-flap span as the trim-flap chord was increased.

The variation of trim-flap chord had no effect on the net lift produced by a multiple-flap system, because the rearward shift in center of pressure of the local flap loads with decreasing flap chord was offset by the decrease in moment arm that resulted from the required increase in trim-flap span. Varying the trim-flap chord thus caused little or no variation in the effective moment arm of trim flaps for a given pitching moment; the lift lost by these flaps therefore was, for practical purposes, independent of flap chord. If the primary consideration is the span available for control surfaces, large-chord trim flaps would generally be desirable.

The data of figure 8 indicate that, for the wing conditions specified, a lift flap of 0.215b is self-trimming. Flap spans smaller than 0.215b create stalling moments and consequently require trim flaps that are deflected downward.

Effect of Flap Deflection

The effect of flap deflection upon the net lift increments produced by flaps is a function only of the resulting increase in angle of attack $\Delta\alpha$ because the center of pressure of flap loads was found to be almost independent of flap deflection. The variation of net lift increment with flap deflection is shown in figure 9. These data indicate that the largest practicable flap deflection should be used. For split flaps on thin to moderately thick wings, however, there is little increase in $\Delta\alpha$ and hence in net lift for deflections greater than 60°.

Effect of Sweepback

The results of the calculations made for a range of sweepback angles from 10° to 30° are shown in figure 10. Arrangements of self-trimming flaps are not possible for a wing of zero sweepback, because the resultant centroid of flap loads would act behind the wing aerodynamic center and would not satisfy the requirements of zero pitching moment. Sweepback of the wing, however, so shifts the wing aerodynamic center and the wing sections that the lift-flap centroids of incremental load move forward

relative to the wing aerodynamic center and trim-flap centroids move rearward. As the sweepback angle is increased, therefore, one angle is reached at which the center of pressure of flap loads at the wing center line acts at the same fore-and-aft location as the wing aerodynamic center. This sweepback angle is the minimum angle at which self-trimming flaps may be employed. For the particular wing used in the calculations shown in figure 10, this minimum sweepback angle was 14.5°. Further increase in sweepback angle allows the use of larger-span self-trimming flaps and results in sizable net lift coefficients. The following values taken from figure 10 illustrate the rapid rise of net lift coefficient with sweepback angle:

Sweepback angle, (deg)	Net lift increase, $\left(\Delta C_{L_f}\right)_{net}$	Allowable span for self-trimming flap (percent b)
14.5	0	0
16	.19	8
18	.32	16
20	.45	22
25	.60	33
30	.74	42

These values do not include consideration of the effect of sweepback on the stalling and hence on the maximum lift characteristics of an airplane.

The results presented in figure 10 indicate that net lift increments larger than those attainable from self-trimming flaps may be obtained by coupling the lift flaps with trim flaps located at the tips - particularly if the sweepback angle is small. Use of these multiple-flap systems provides the means of obtaining lift increments from trailing-edge flaps for sweepback angles below the minimum angle for self-trimming flaps. A multiple-flap system, however, provides a smaller lift increase per unit span utilized by the flaps than the self-trimming flap.

The lift increase due to increase in allowable flap span with increased sweepback angle is not all gain inasmuch as an increase in sweepback angle also reduces the lift-curve slope and shifts the wing stall to the wing tips. The theoretical analysis of reference 11 indicates

that the loss in lift-curve slope varies approximately as the cosine of the angle of sweepback. As a rough approximation, the maximum lift coefficient of a wing also was assumed to vary as the cosine of the angle of sweepback. Figure 11 presents available wind-tunnel force-test data concerning the reduction of $C_{L_{max}}$ due to sweepback angle. In addition to reducing maximum lift, wing-tip stalling may also produce undesirable longitudinal and lateral stability characteristics. Some feature that would eliminate or reduce the effects of the wing-tip stalling should therefore be incorporated in the design of the wing. Wing-tip slots or washout has already proved to be beneficial in this respect. The results of reference 15 indicate that as much as 30 percent of the lift loss due to sweepback may be recovered by washout. Part of the lift loss due to sweepback will probably be recovered on most tailless airplanes, inasmuch as some washout is usually used.

Effect of Aspect Ratio

Results of the aspect-ratio calculations are presented in figure 12 and indicate that sizable net lift increments cannot be obtained, even for wings with large sweepback angles, when the aspect ratio is small (less than 5). Increasing the aspect ratio above 5 resulted in a rapid increase in the allowable flap span. At high aspect ratios (about 10), lift increments comparable with those for conventional airplanes were calculated for tailless airplanes. These trends are illustrated in the following table:

A	$\left(\Delta C_{L_f}\right)_{net}$	Allowable span for self-trimming flap (percent c)
5.5	0	0
6.0	.18	9
8.0	.54	29
10.0	.82	45
12.0	1.00	56
14.0	1.10	61

The results of figure 12 show also that the efficiency of the trim flap in neutralizing pitching moments without

undue loss in lift decreases with increased aspect ratio and that use of this flap system is not warranted at high aspect ratios. For high aspect ratios, the single self-trimming flap occupies almost all or all the span that can be allowed for flaps without reducing the span necessary for lateral control and, as previously shown, the self-trimming flap yields higher lift increments than the multiple-flap system of equal span.

The aspect-ratio calculations explain the difficulties encountered in trying to obtain high lifts on tailless fighter airplanes. In an investigation in the NACA free-flight tunnel (unpublished data) of a model of a tailless fighter airplane with aspect ratio of 5.1 and with sweepback angles of 24° and 34°, a maximum increase in the net incremental maximum lift coefficient of only 0.11 was obtained with the best of a large variety of flap systems operating under trimmed-flight conditions. The present calculations show that a lift increment nine times this value would be obtained for a similar airplane having twice the aspect ratio. Trailing-edge-flap lift increments approaching those of flapped conventional airplanes thus seem to be obtainable only on tailless airplanes of fairly high aspect ratio.

The results of the aspect-ratio calculations can be explained by consideration of the action of increased aspect ratio upon the aerodynamic center of a swept-back wing. As previously shown by equation (3), the fore-and-aft location of the wing aerodynamic center is defined by the relation $x_{a.c.} = Hb \tan \Lambda$. For a given wing area, an increase in aspect ratio increases both H and b and consequently results in a rearward shift in the wing aerodynamic center. This action results in a reduction of the diving moments created by flaps and hence allows extension of the lift-flap spans for a given pitching moment. This fact explains the rapid increase in net lift increment with increased aspect ratio. The moment arms of the outboard sections are similarly reduced with increase in aspect ratio and a greater lift loss is thereby incurred for a given required trimming moment. Inspection of the formula $x_{a.c.} = Hb \tan \Lambda$ indicates that the largest effect of aspect ratio occurs when the sweepback angle is large. High lifts are most readily attained, therefore, for relatively large values of both aspect ratio and sweepback angle.

Effect of Taper Ratio

Results of taper-ratio calculations are presented in figure 13, which shows that an increase in taper ratio allows a sizable increase in the net lift increments of self-trimming flaps only when the taper ratio is initially small. For values of taper ratio higher than 0.5, the gain in net lift increment with increased taper ratio is negligible. For the airplane investigated, increasing the taper ratio from 0.25 to 0.50 increased from 0.42 to 0.66 the allowable net lift increment obtainable by self-trimming flaps. A further increase in taper ratio to 1.0 increased the net lift increment to only 0.70. The results of figure 13 also show a successive decrease in trim-flap efficiency as the taper ratio is increased from small values to 0.5. Above 0.5, however, a slight gain in the trim-flap efficiency is afforded by increased taper ratio. This value seems to indicate the point at which the direct effect of increasing the tip chord offsets the decreased moments of the outboard sections caused by the shift of the aerodynamic center.

An increase in taper ratio results in a rearward shift of the wing aerodynamic center. It would thus be expected that, as the taper ratio is increased, the flap moments would be reduced and, therefore, that self-trimming flaps of larger span could be used. Increasing the taper ratio of a given wing, however, causes also a reduction in the chords of the inboard sections and an increase in the chords of the outboard sections. The gain in lift-flap span allowable, due to the aerodynamic-center movement with increased taper ratio, is thus increasingly opposed by the decreasing chord of the inboard sections. In a similar manner, the decrease in trim-flap efficiency caused by shortened moment arms with increased taper ratio is offset by the increase in chord of the outboard sections.

In addition to the favorable effect of moderate taper ratio (about 0.5) upon the lift produced by flaps on tailless airplanes, further advantages are realized by taper ratios of this order. The wing-tip stalling characteristics are more satisfactory for moderately tapered wings than for highly tapered wings. Moderately tapered designs therefore allow use of larger angles of sweepback before sizable losses in lift are incurred. Reference 17, in addition, shows that moderately tapered wing designs (taper ratio between 0.33 to 0.5) are structurally most efficient.

Lift Loss in Obtaining Longitudinal Trim

The calculations made to determine the effect of the parameters on the lift and pitching-moment characteristics of the trim flaps are also applicable to the study of the lift loss incurred in balancing the stability moments of the airplane. The effects of varying the trim-flap chord are presented in figure 8, which shows that no appreciable change in lift occurs with varying the trim-flap chord for a given pitching moment. A study of the effect of taper ratio also indicated that this effect on the lift loss due to trim would be negligible. The effect of taper ratio on the lift loss incurred in trimming a moment of 0.10 about the aerodynamic center of a wing with sweepback angle of 20° and aspect ratio of 7.3 is shown in the following table:

λ (deg)	ΔC_{L_T}
0.25	-0.23
.50	-.21
1.00	-.18

The effects of aspect ratio and sweepback on the lift loss resulting from the additional trim required for longitudinal control are summarized in figure 14. These data indicate that large values of aspect ratio and sweepback angle are required to minimize the lift loss resulting from the additional trim requirements.

The data of figure 14 may be applied to trim controls using different chords or deflections or to control surfaces other than the plain split flap, if the centroid of incremental flap load is located at approximately the same chordwise station as that of the split flap. An incremental angle of attack $\Delta\alpha = 8.4^{\circ}$ was used to make the calculations required for figure 14. In order to convert the data of figure 14 to other flap chords, flap deflections, or airfoil sections, the scales of the graphs can be multiplied by the ratio of incremental angles of attack.

Estimation of Maximum Lift Coefficient of Flapped Tailless Airplanes in Trimmed Flight

During the course of the investigation, it was recognized that the data obtained in the analytical and empirical studies could be employed to give an indication of the maximum lift coefficient attainable with trailing-edge split flaps on tailless airplanes.

The maximum lift coefficient of a tailless airplane in trimmed flight with flaps deflected may be expressed in summary form as

$$\left(C_{L_{max}}\right)_{tr} = \left(C_{L_{max}}\right)_{w} + \Delta C_{L_f} - \text{Losses}$$

The lift losses of a tailless airplane may be considered to originate from three sources - the loss due to the longitudinal control surface, the loss in maximum lift induced by sweepback, and the loss due to change of wing stalling characteristics when the flaps and longitudinal control surface are deflected. The first two sources have already been discussed. The loss due to trim is given in figure 14, and the loss induced by sweepback can be assumed to vary roughly as the cosine of the angle of sweepback. The change in stalling characteristics of a wing, when the flaps and longitudinal control surface are deflected, may change the angle of attack at which the maximum lift occurs and may change also the characteristics of the nonlinear portion of the lift curve. These effects may cause the increase in maximum lift produced by flaps to vary from the value of the lift increase produced by flaps at a wing angle of attack of 10°. Because stall prediction is uncertain and because of the complex nature of the problem, no attempt has been made in the present investigation to analyze accurately the factors affecting the flapped-wing stall. A statistical study, however, was made concerning the ratio of incremental maximum lift coefficient produced by flaps $\left(\Delta C_{L_f}\right)_{max}$ to the incremental lift coefficient produced by flaps at an angle of attack of 10° $\left(\Delta C_{L_f}\right)_{\alpha=10^\circ}$. These data were obtained from available wind-tunnel force-test data and are presented in figure 15.

These data indicate that a ratio of $\left(\Delta C_{L_f}\right)_{max} / \left(\Delta C_{L_f}\right)_{\alpha=10^\circ}$ of 0.9 would be a good mean value. An indication of the maximum trim lift coefficient can now be obtained from the relationship

$$\left(C_{L_{max}}\right)_{tr} = \left(C_{L_{max_{bw}}} + 0.9\ \Delta C_{L_f}\right) \cos \Lambda - \Delta C_{L_p} \qquad (4)$$

where ΔC_{L_p} corresponds to the lift lost by a control surface that yields a pitching-moment coefficient about the aerodynamic center equal to $h\left(C_{L_{max}}\right)_{tr}$ and may be obtained from figure 14. It should be noted that ΔC_{L_p} is a function of $\left(C_{L_{max}}\right)_{tr}$ and hence equation (4) has to be solved by the trial-and-error method.

High-Lift Design for Tailless Airplanes

The present analysis has indicated that by proper design a tailless airplane may obtain maximum lift coefficients comparable with those of conventional aircraft. A design that combines the features shown by the present results to be favorable for achieving high values of maximum lift coefficient is shown in figure 16. With this design, it is estimated that a maximum lift coefficient of 2.0 in trimmed flight with a 5-percent static margin may be obtained.

CONCLUSIONS

The conclusions drawn from an analysis of factors affecting the net lift increment obtainable from trailing-edge split flaps on tailless airplanes in trimmed flight follow:

1. The maximum lift coefficient of tailless airplanes may be of the order of 2.0 or greater for reasonably large values of aspect ratio (10) and sweepback angle (20°). Low values of aspect ratio (5) and sweepback angle (10°)

will limit the incremental lift obtained with flaps to small values and will also result in excessive lift losses due to longitudinal control.

2. The highest net lift increments will be obtained from flaps on wing designs that allow use of single inboard self-trimming lift flaps which occupy all the wing span not taken up by other control surfaces.

3. For wing designs that limit the span of the inboard self-trimming lift flap to a fraction of the span otherwise allowable, further increases in net lift may be obtained by use of a multiple-flap system consisting of an inboard lift flap deflected downward and an outboard trim flap deflected upward.

4. Excessive taper will reduce the net lift obtainable from flaps on tailless airplanes. Taper ratios of 0.5 or greater are recommended for tailless-airplane designs.

5. Increased flap chord and deflection will lead to increased net lift increments due to flaps.

6. For wing-tip elevators, aspect ratio and sweepback are the controlling factors in minimizing the lift loss for obtaining trim of a given pitching moment. Taper ratio and elevator chord have little effect on the trim loss.

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National Advisory Committee for Aeronautics
Langley Field, Va.,

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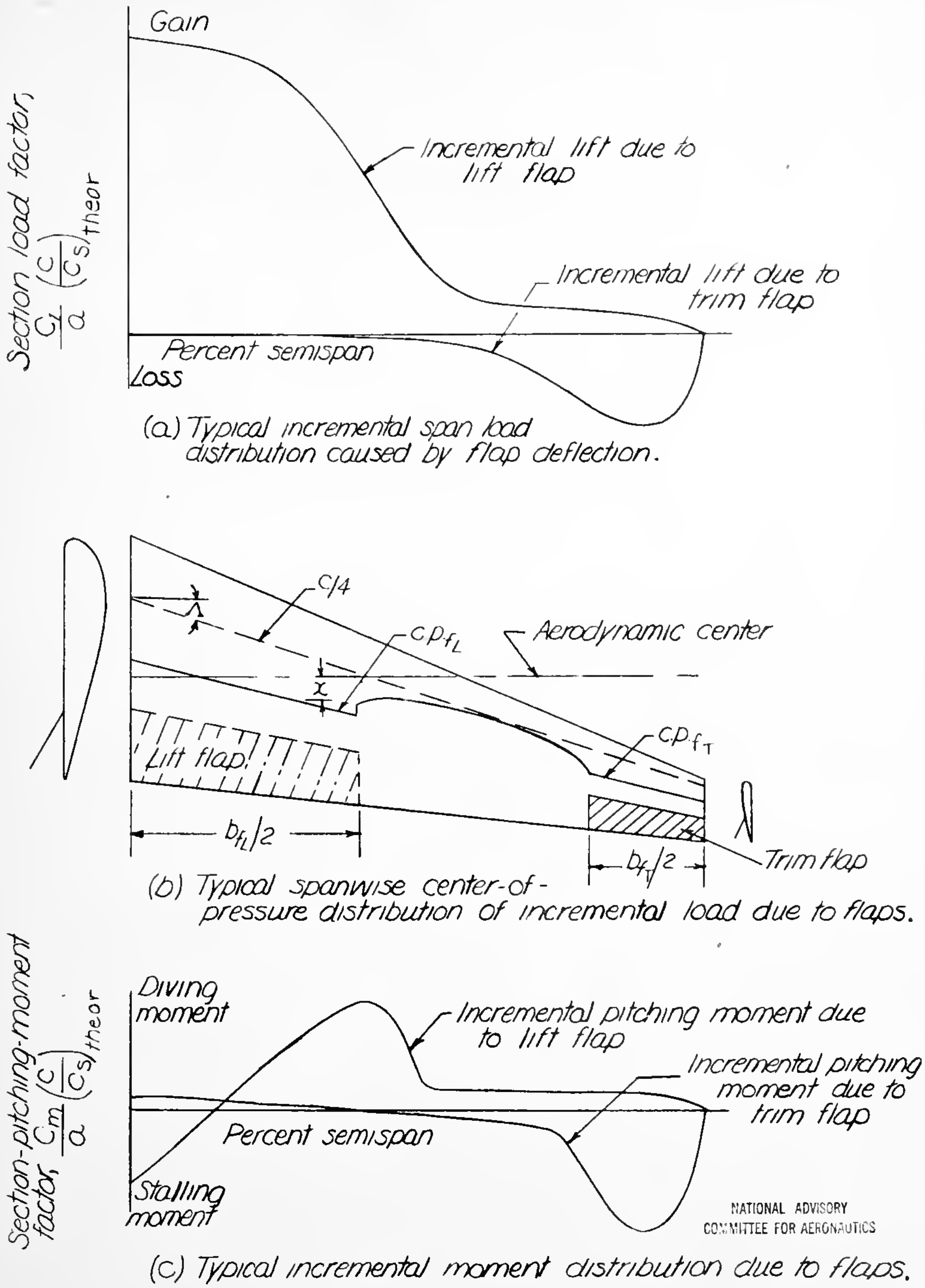


Figure 1.- Sample lift, center-of-pressure, and pitching-moment distributions due to partial-span flaps on a swept-back wing.

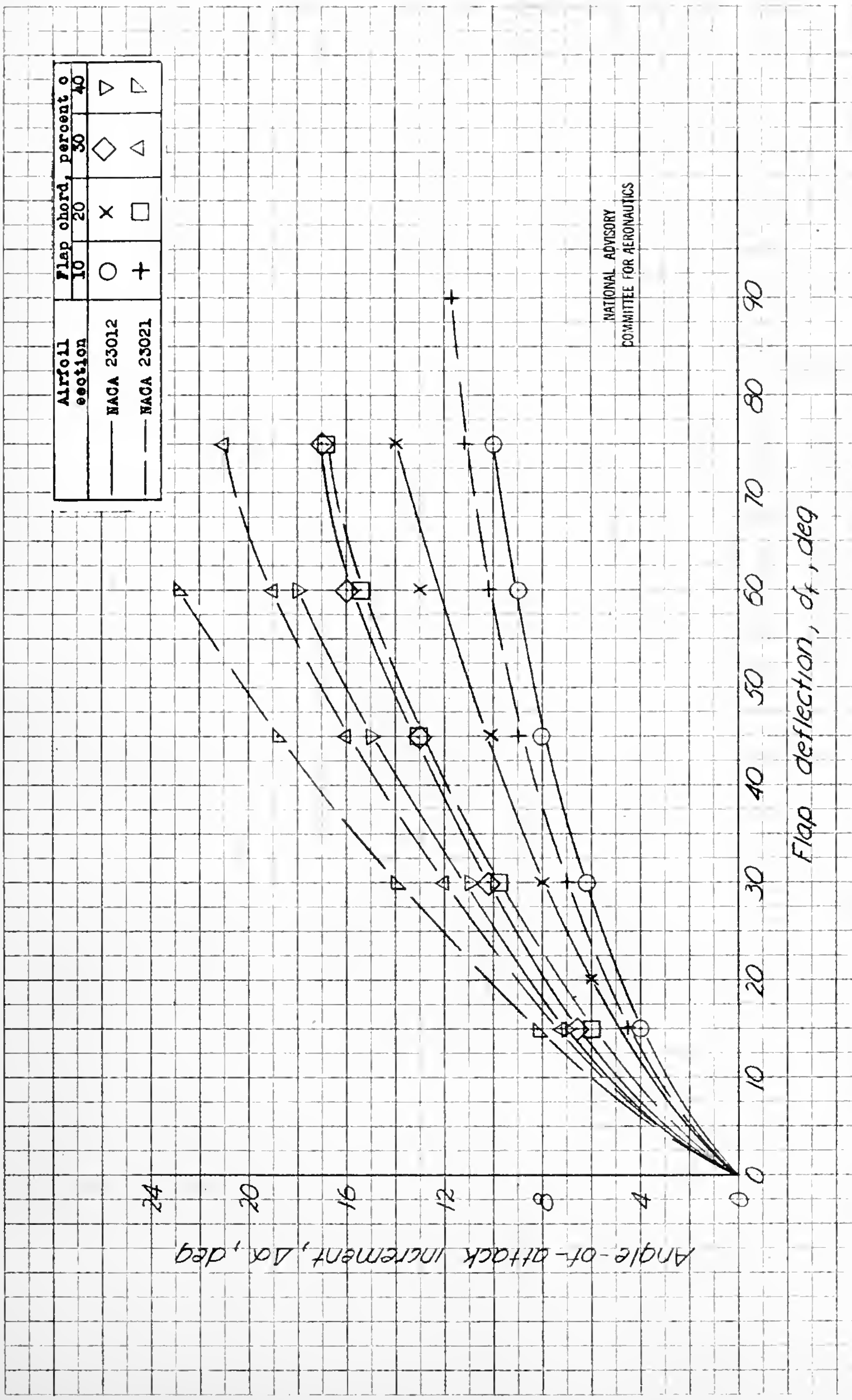


Figure 2.- Effectiveness' of split flaps of various chords at $C_L = 1.0$. Data from reference 3.

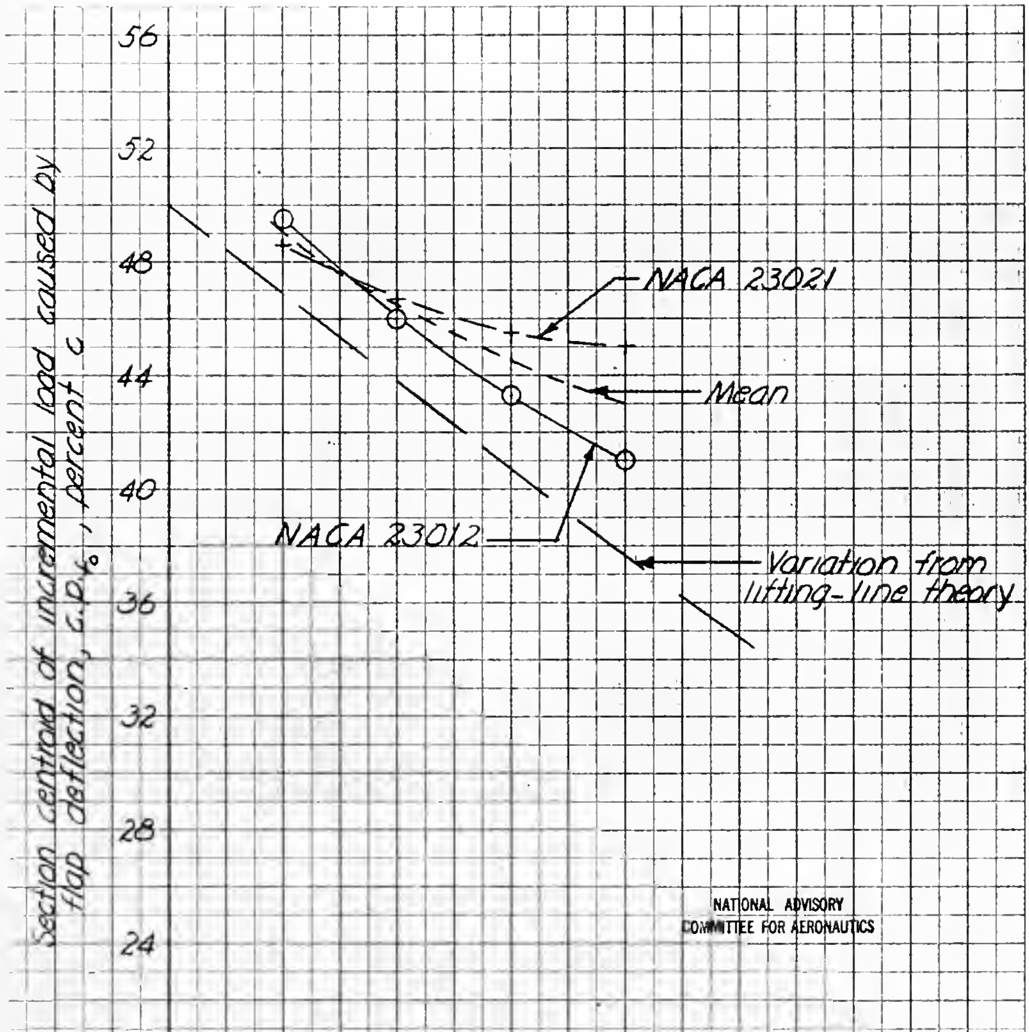


Figure 3.- Effect of flap chord and airfoil thickness upon centroid of load caused by flap deflection. $\alpha=10°$. Data obtained from reference 3.

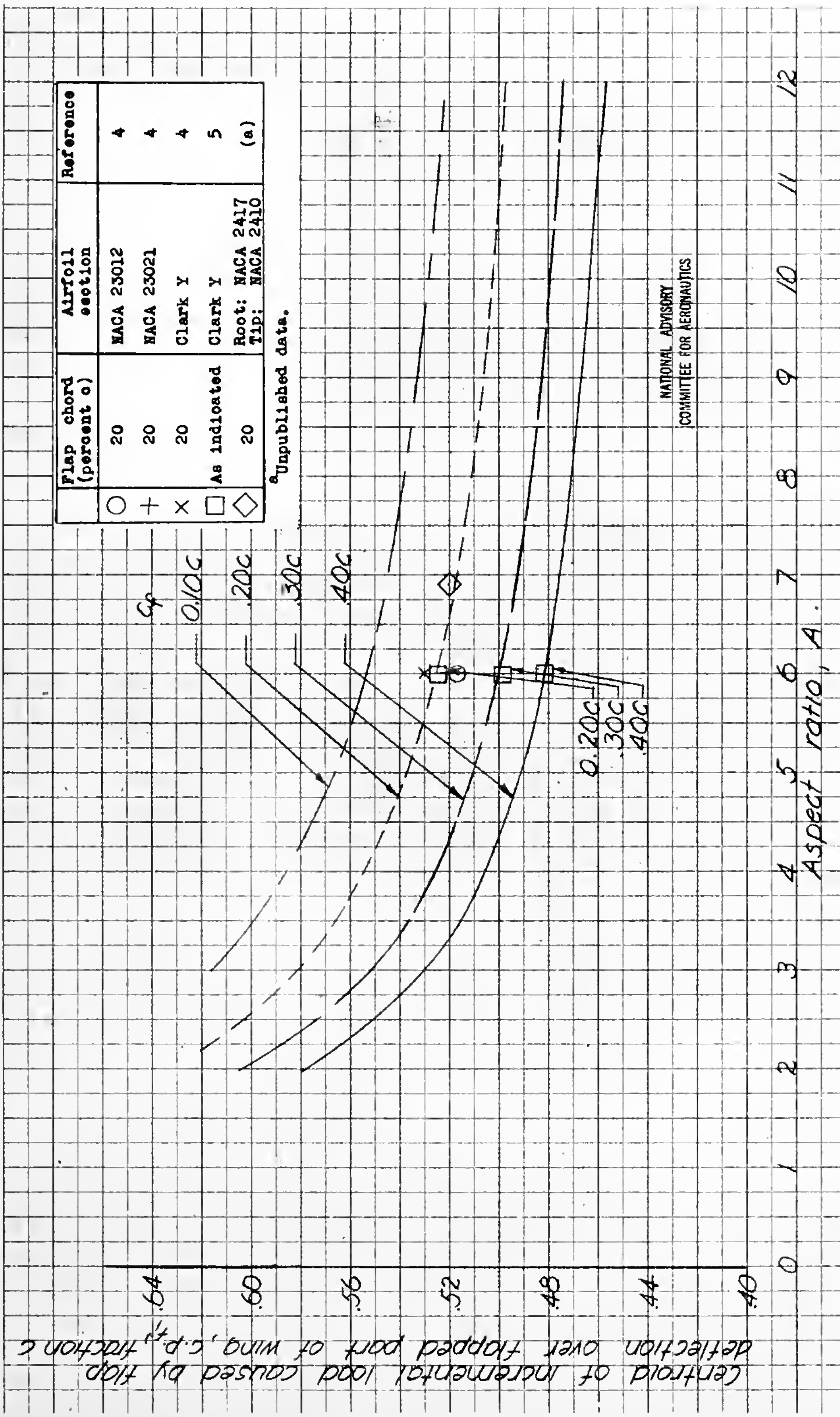


Figure 4.- Effect of aspect ratio upon the center of pressure of load caused by flap deflection for various-chord split flaps. $\alpha = 10°$, $t/c = 0.16$.
$c.p._{f_1} = \frac{a_0}{a}(c.p._{f_0} - 0.25) + 0.25$.

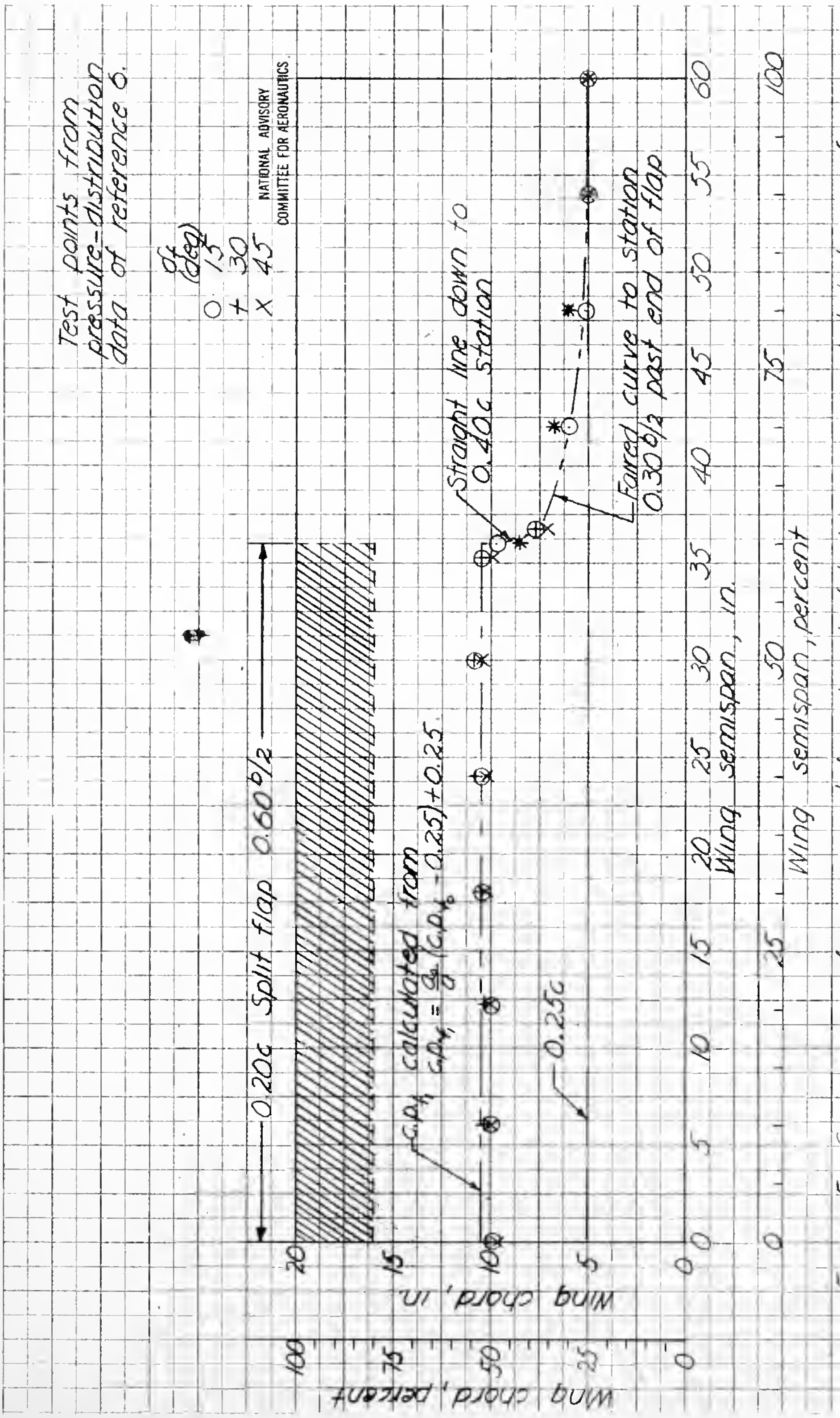
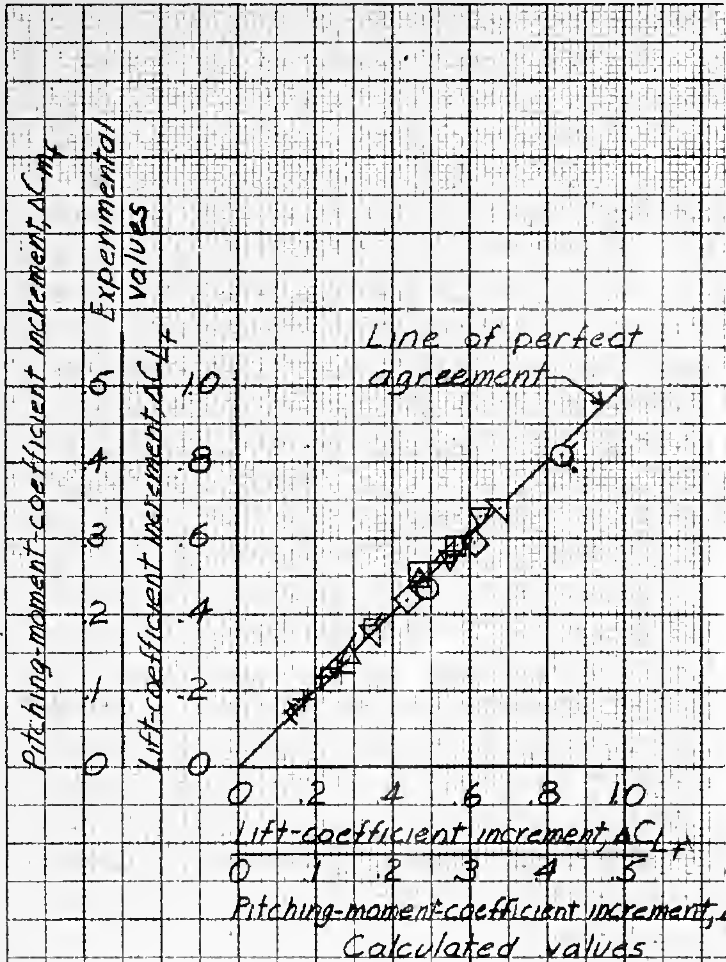


Figure 5.- Comparison of experimental and calculated spanwise distributions of centroid of incremental load caused by partial-span flap. A=6; λ=1.0.



ΔC_{L_f}	ΔC_{m_f}	Plan form	A	λ	Λ (deg)	c_f (percent c_w)	Flap span (percent b/2)	δ_f (deg)	Flap type (a)	Reference
○ + ×	○ + ×	NACA 65(318)-019	7.3	0.25	22	23 22 25	34 34 20	60 20 50	[1]Split [2]Plain [3]Plain	(b)
□	□	Clark Y	3.0	0.33	27	16.7 to 50	100	15	Plain	7
◇	◇	Root: NACA 23015 Tip: NACA 23009	7.0	Eff. 0.50	16	20	53	45	Split	8
△ ▽	△ ▽	Root: M45-018 Tip: M45-012	5.1	0.17	35	20 20	47 76	40 40	[1]Split [2]Split	(b)
◸	◸	NACA 5-10-16	10.0	0.20	0	20	50	20	Plain	2
◁	◁	Clark Y	6.0	0.60	2.5	15	70	60	Split	9
◺	◺	Root: NACA M9 Tip: NACA 0009	7.0	0.32	-15	20	62	45	Split	(b)

[a]Superscripts indicate flaps numbered as shown in sketches.
[b]Unpublished data.

Figure 6.- Comparison of experimental and calculated increments of lift and pitching-moment coefficients due to flap deflection for various wing-flap combinations. $\alpha = 10°$

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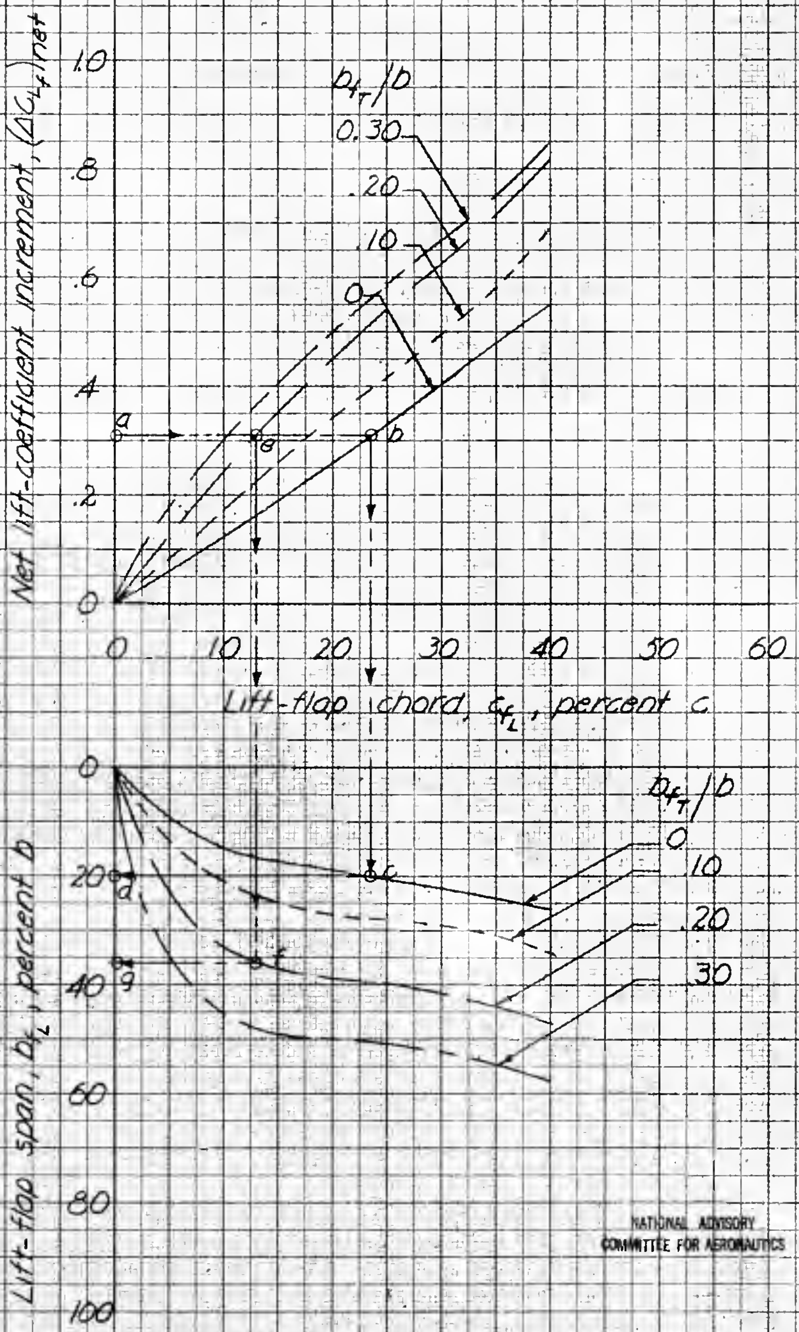

Figure 7.- Effect of lift-flap chord on net increment of lift coefficient and lift-flap span for various trim-flap spans. $\Delta C_{m_{a.c.}} = 0$; $\Lambda = 20°$; $A = 7.35$; $\delta_{f_L} = 60°$; $\delta_{f_T} = 60°$; $c_{f_L} = c_{f_T}$.

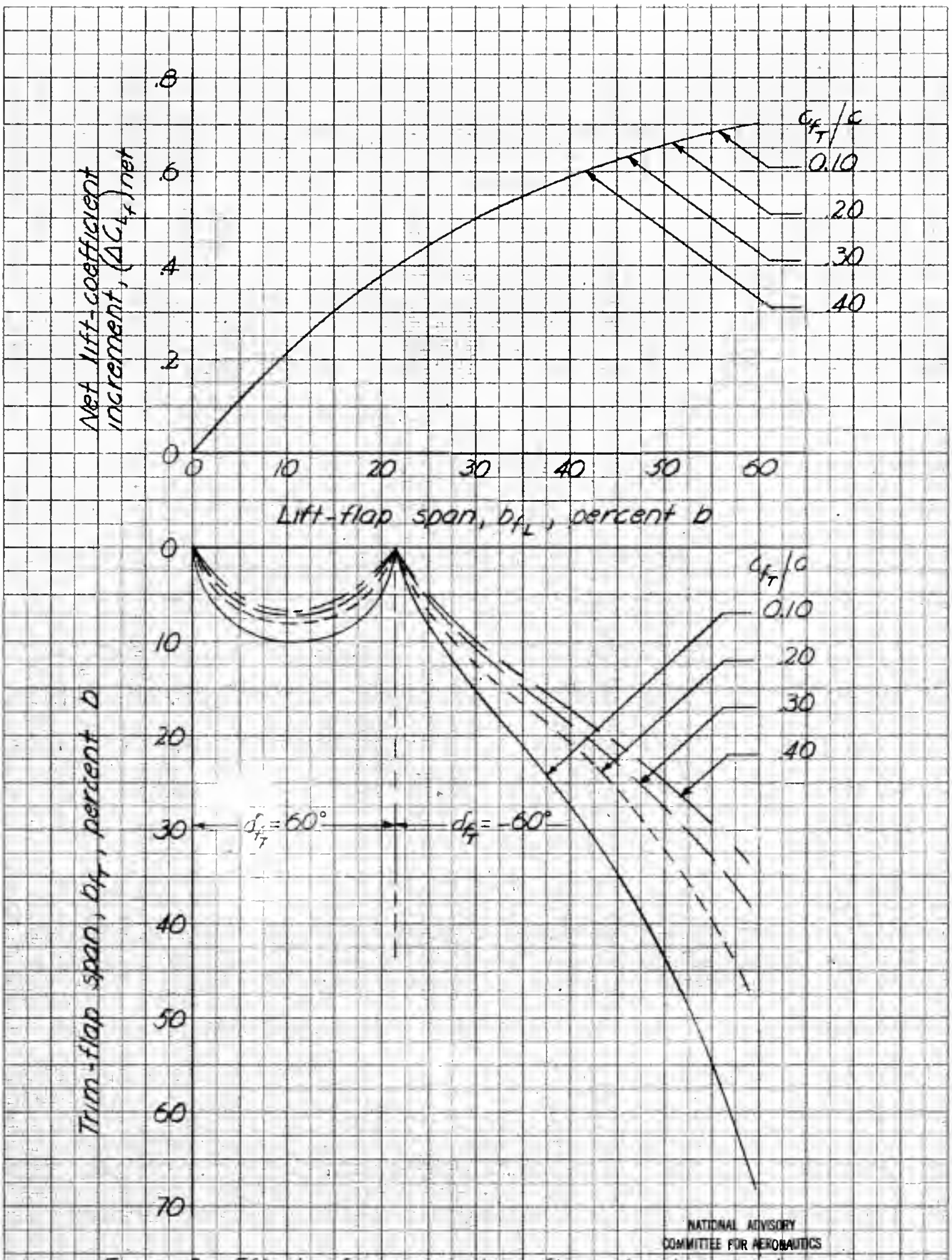

Figure 8.- Effect of varying trim-flap chord on net increment of lift coefficient and trim-flap span. $\Delta C_{m_{a.c.}} = 0$; $\Lambda = 20°$; $A = 7.35$; $\lambda = 0.25$; $c_{f_L} = 0.30c$; $\delta_{f_L} = 60°$.

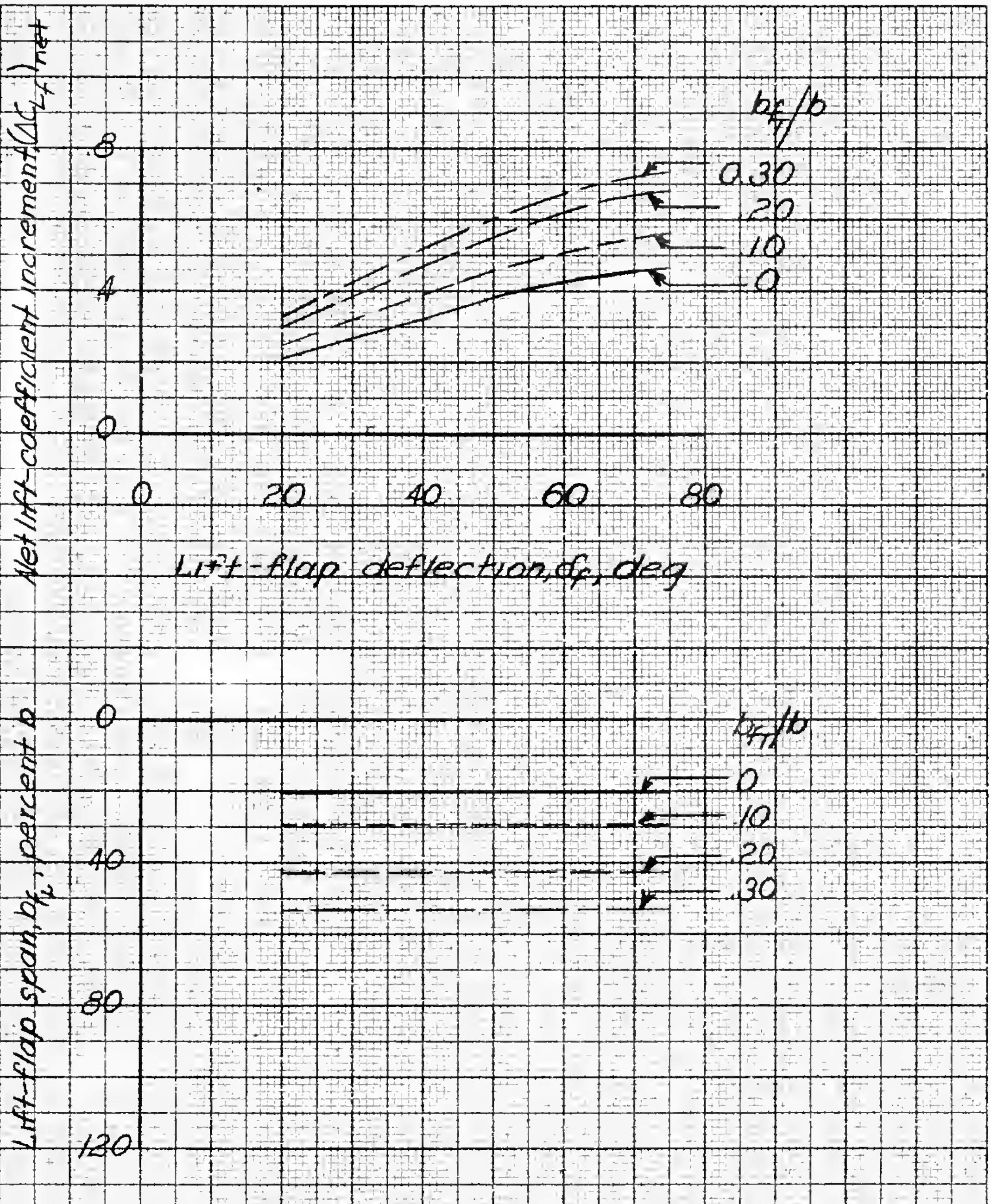

Figure 9.- Effect of flap deflection on net increment of lift coefficient and lift-flap span for various trim-flap spans. $\Delta C_{m_{a.c.}} = 0$; $\Lambda = 20°$; $A = 7.35$; $c_{f_L} = 0.30c$; $c_{f_T} = 0.30c$; $\delta_{f_T} = \delta_{f_L}$; $\lambda = 0.25$.

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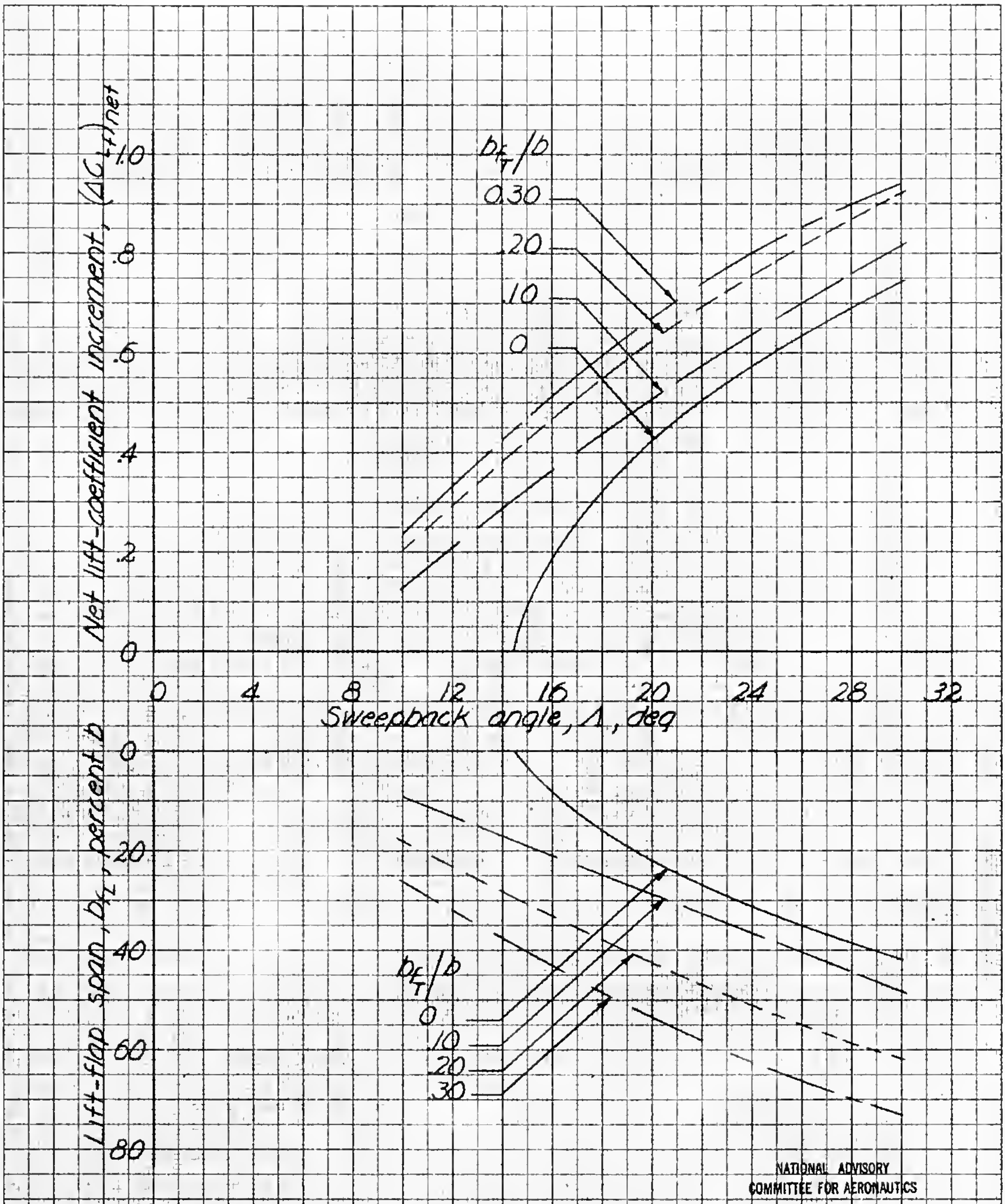
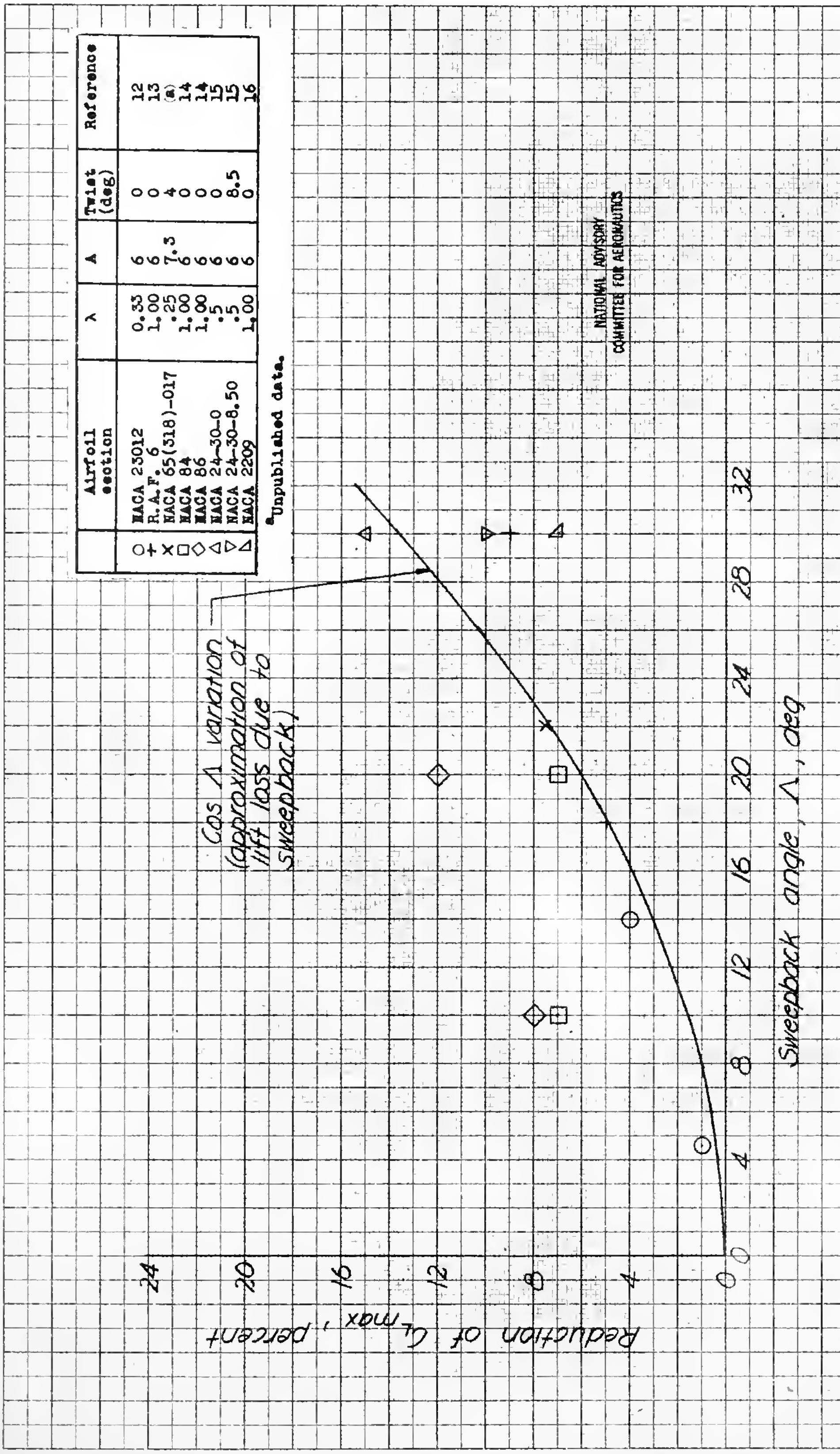


Figure 10.– Effect of sweepback on net increment of lift coefficient and lift-flap span for various trim-flap spans. $\Delta C_{m_{a.c.}} = 0°$; $A = 7.35$; $\lambda = 0.25$; $c_{f_L} = 0.30c$; $c_{f_T} = 0.30c$; $\delta_{f_L} = 60°$; $\delta_{f_T} = 60°$.



	Airfoil section	λ	A	Twist (deg)	Reference
○	NACA 23012	0.33	6	0	12
+	R.A.F. 6	1.00	6	0	13
×	NACA 65(318)-017	.25	7.3	4	(a)
□	NACA 84	1.00	6	0	14
◇	NACA 86	1.00	6	0	14
◁	NACA 24-30-0	.5	6	0	15
▷	NACA 24-30-8.50	.5	6	8.5	15
△	NACA 2209	1.00	6	0	16

[a]Unpublished data.

Figure 11.- Loss in maximum lift coefficient due to sweepback angle

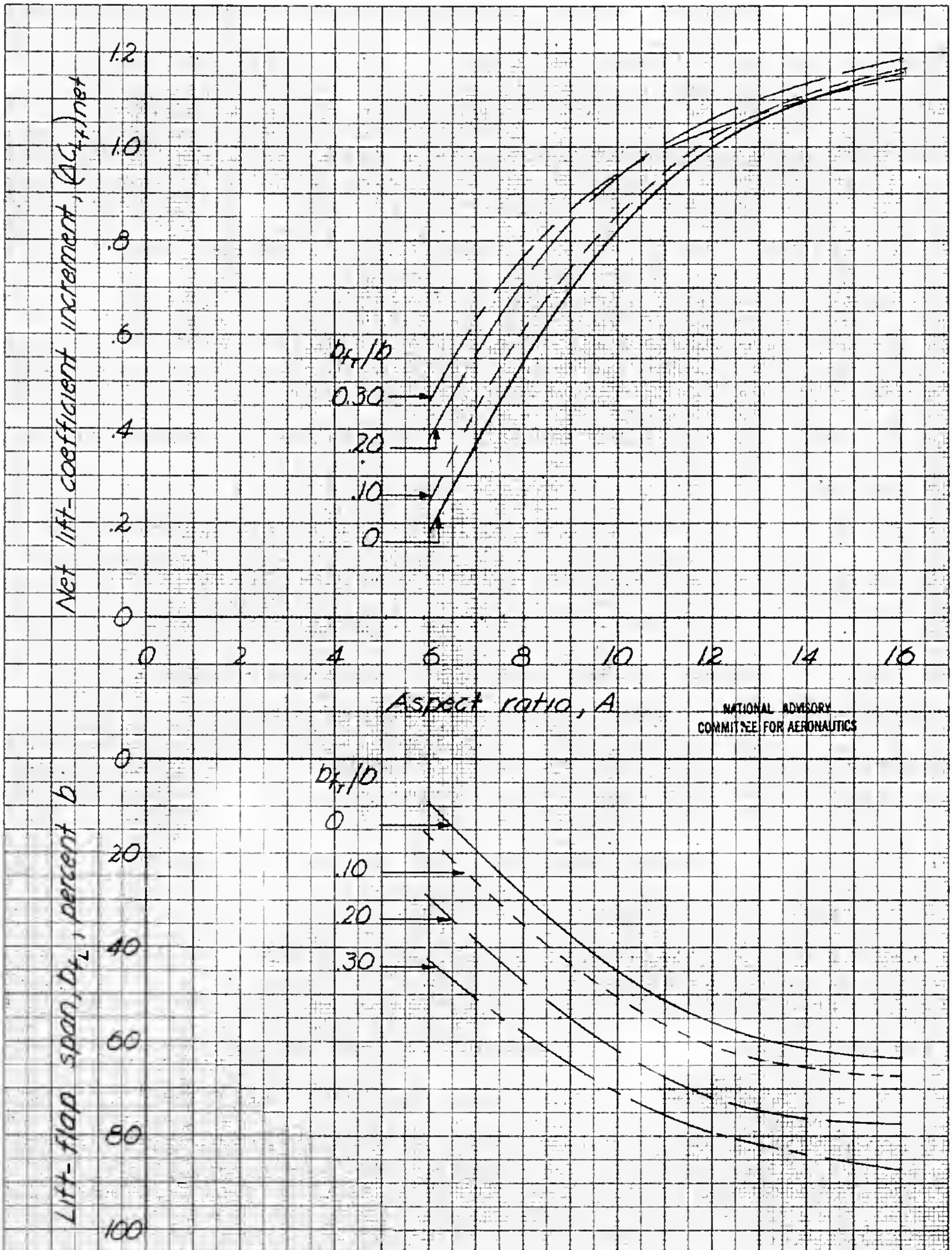


Figure 12.- Effect of aspect ratio on net increment of lift coefficient and lift-flap span for various trim-flap spans. $\Delta C_{m_{a.c.}} = 0$; $\Lambda = 20°$; $\lambda = 0.25$; $c_{f_L} = 0.30c$; $c_{f_T} = 0.30c$; $\delta_{f_L} = 60°$; $\delta_{f_T} = 60°$.

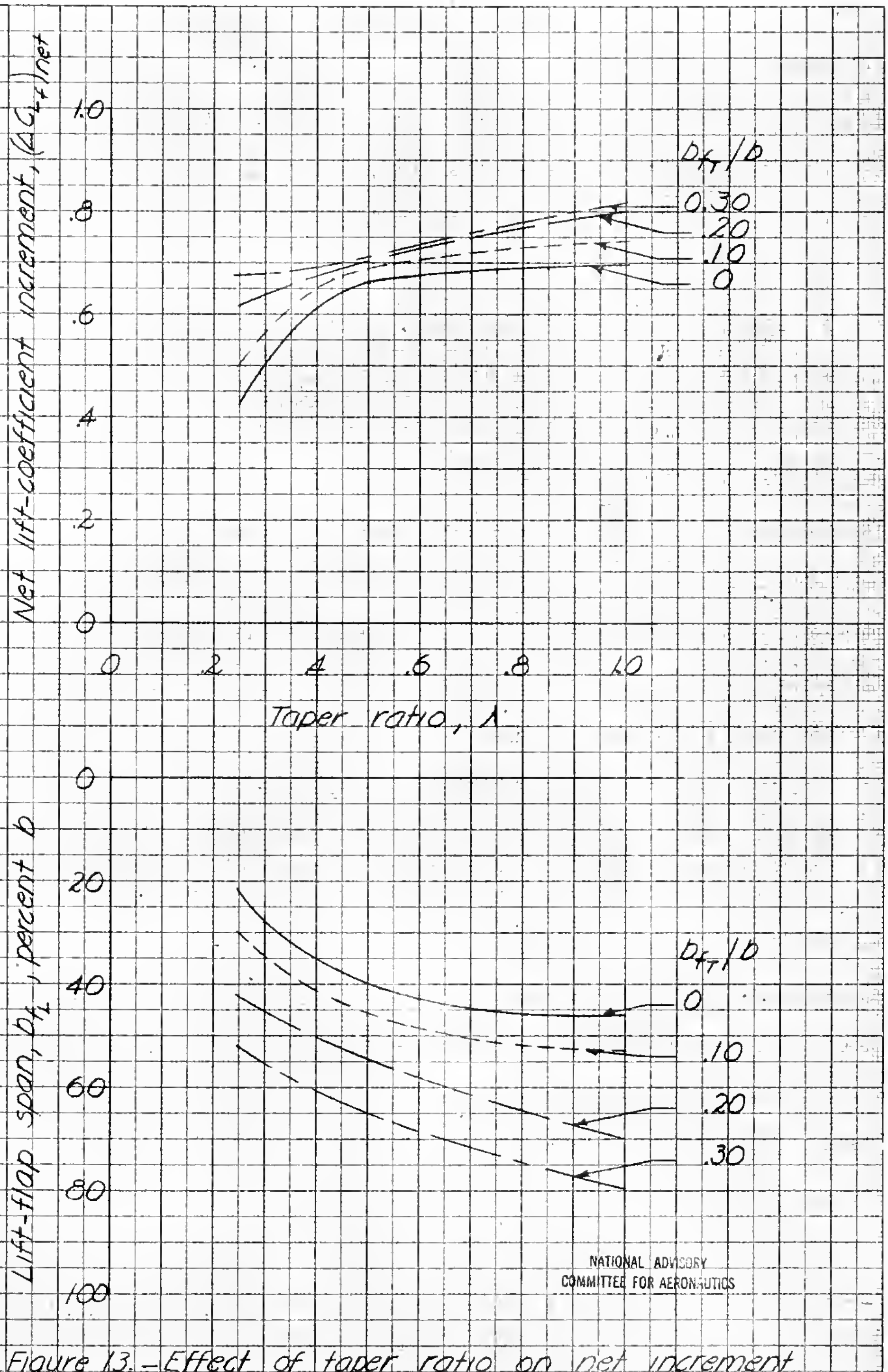


Figure 13.– Effect of taper ratio on net increment of lift coefficient and lift-flap span for various trim-flap spans. $\Delta C_{m_{a.c.}} = 0$; $\Lambda = 20°$; $A = 7.35$; $c_{f_L} = 0.30c$; $c_{f_T} = 0.30c$; $\delta_{f_L} = 60°$; $\delta_{f_T} = 60°$.

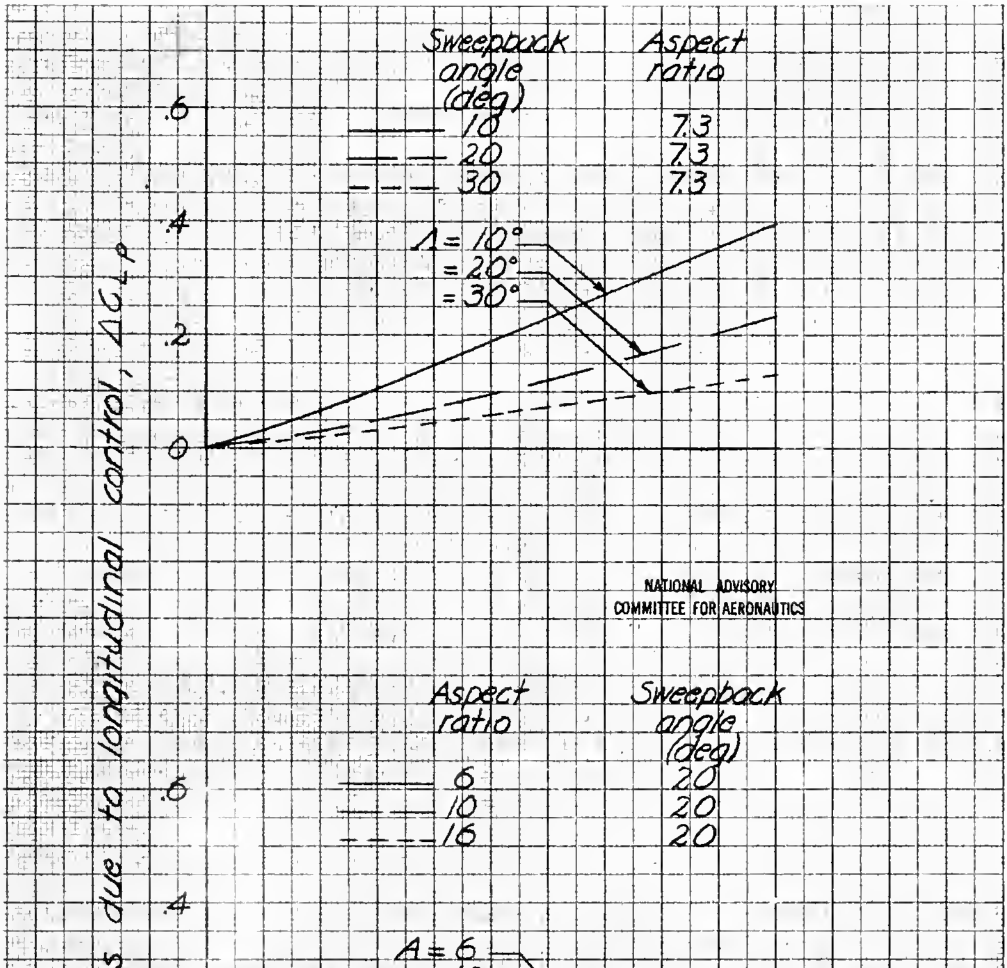


Figure 14.– Loss in lift coefficient due to longitudinal controls extending inboard from wing tips. $c_p = 0.30c$; $\delta_p = -21°$; $\lambda = 0.25$.

Case	Airfoil section	b_f (fraction b)	c_f (percent c)	δ_f (deg)	A	λ	Reference
1	Clark Y	0.5	15	60	6	0.2	9
2		.7				.2	
3		1.0				.2	
4		.5				.6	
5		.7				.6	
6		1.0				.6	
7	NACA 22 series	0.82	20	10	6	1.0	18
8				20			
9				40			
10				60			
11				75			
12	NACA 23009	1.0	20	75	6	1.0	19
13	NACA 43009						
14	NACA 63009						
15	NACA 23012						
16	NACA 43012						
17	NACA 23015						
18	NACA 23021						
[a]19	NACA 23012	0.60	20	60	6	0.33	12
[a]20							
[a]21							

[a]Wing-fuselage combination: fuselage, circular; sweepback angle, 4.75°; high wing, midwing, and low wing, respectively, represented.

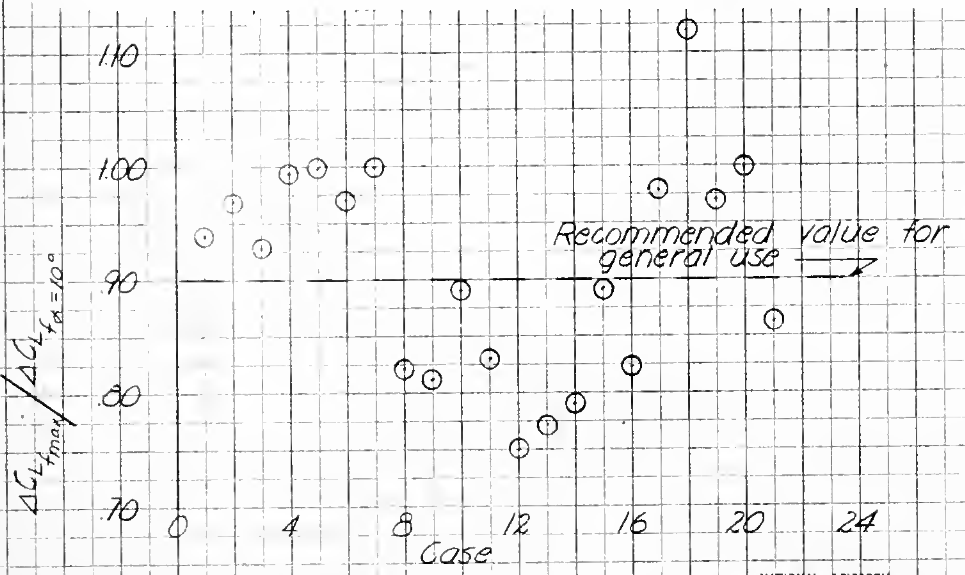


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Figure 15.- Ratio of incremental lift coefficient caused by flaps at maximum lift to that at $\alpha = 10°$.

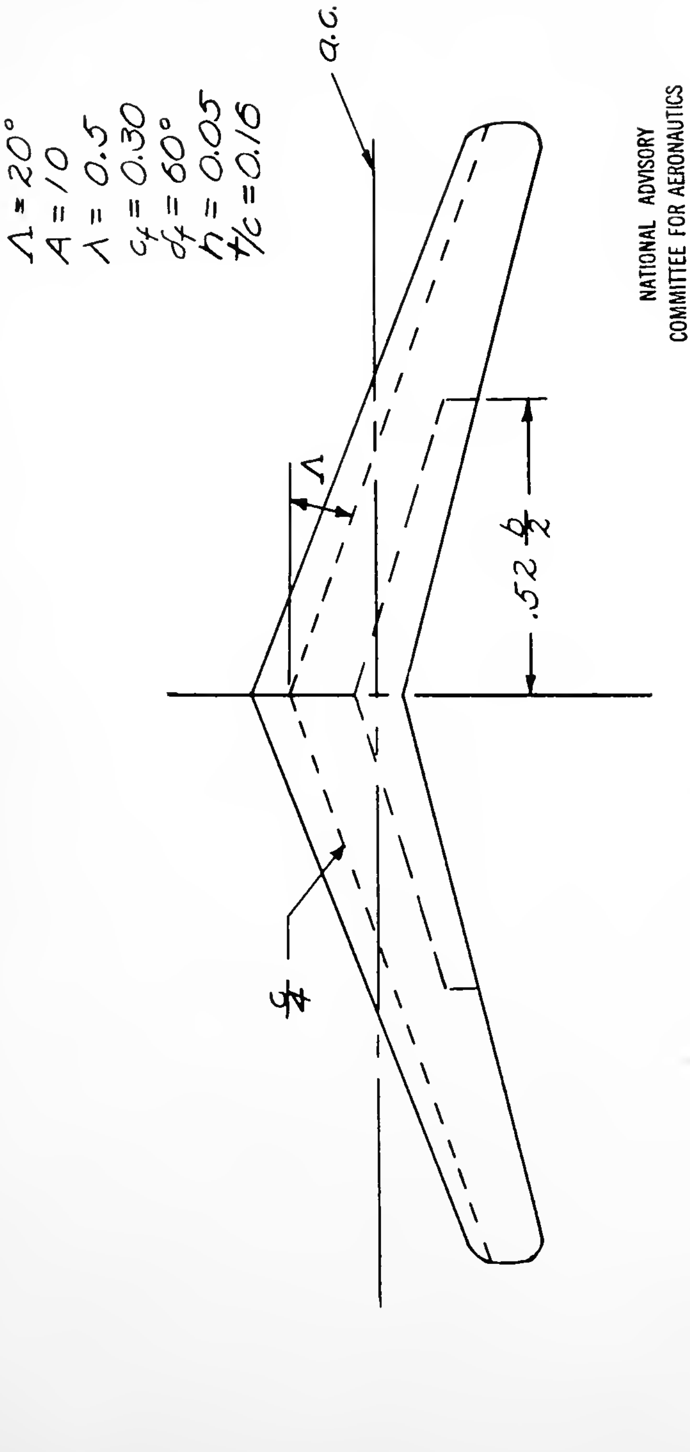

Figure 16.- Example of moderately tapered tailless airplane of high aspect ratio equipped with a self-trimming flap, estimated to give a maximum lift coefficient of 2.0 in trimmed flight.

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